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THE ION R

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THE ION ROCKET

- Hungary -

[Following is the translation of an article by Erno Nagy in Fizikai Szemle (Physics Review), Vol XI, No 1 Budapest, 1961, pages 3-12.]

1) Some Introductory Remarks.

Even the physicists and engineers who deal with common rocket engines and the professor who teaches the principles of rocket motion always encounter the problems pertaining to different unit systems. There are few areas in modern physics and technics where the confusion around the units of mass and force (weight) areas significant as in the field of rocket engines. It is not without reason that a great number of rocket technicians calculate constantly with specific impulses (I_1 or I_{sp}), because by employing these characteristic quantities, the units of weight and not those of mass may be used in the basic equation of rocket motion.

The well-known basic equation of rocket motion is

$$P = \frac{dm}{dt} \cdot w = \dot{m} \cdot w \qquad (1)$$

where P is the thrust (reaction force), $\dot{m}$=dm/dt is mass consumption per second (the mass of the propellant thrusted out in unit time) and w is the (relative) velocity of discharging this mass.

By knowing that m = G/g (the mass is the quotient of the weight and the gravitational acceleration), the former equation may be written in the following form:

$$P = \frac{dG}{dt} \cdot \frac{w}{g} = \dot{G} . I_1 \qquad (2)$$

where already the loss of weight per second (the weight of propellant discharged in unit time), is involved. On the other hand, the quotient w/g represents the specific impulse I_1 mentioned previously. Thereby the introduction of the specific impulse succeeds in relating the equation of the thrust to the weight of the propellant.

The question, why a calculation with propellant weight and not the mass is more practical arises frequently. This question is considered to be even more justified, since in above equation the special earth-level value of g appears, while the mass unrelated to the gravitational acceleration may be considered as invariant quantity.

Even if the originators of above objections are theoretically right, we have to take into consideration that today (and presumably always in designing space researching equipment) the designer, especially the designer of the carrier rocket, is interested not in the masses, but in the weights to be lifted from the earth and to be orbited on a given course. In the course of final calculations we have to return in every case to the weight and the force; therefore, it is practical to modify in advance the expressions derived from the dynamics of a point of changing mass so that in these possibly consistent systems the weight and not the mass is involved. We should repeat that this is a requirement of practical work, but it is by all means rational, for it eases the clarification of the relationships and eliminates the source of error in conversion.

This is all the more necessary since, when turning from the mechanics of common chemical rockets to the electrically accelerating propelling systems, the problems related to measuring units will be multiplied; this, on the other hand, introduces into the calculations constants of the most diverse and frequently the most unusual form.

In the following discussion, we can not arrive at the ideal situation. What we can do is to show in every case the modifications in the conversion of CGS and technical unit systems from each other, and thus illustrate with practical problems how many labyrinths and sources of error there are in dealing with different mixed-unit systems.

2) The Crisis of the Classical Rocket

Before elaborating in detail on the topic of our study, we have to answer one other question: why do we need these propelling systems, which appear today as still exotic?

We must state categorically that these propelling mechanisms are not suitable for solving the problems of present space research. Everything that is being realized in the space around our earth and in the neighborhood of the planets of our solar

system and that which will come to realization in the following years will still remain within the possibilities of the chemical rockets; furthermore, the chemical rockets even provide a better solution than any other more advanced propelling mechanism.

We shall not disregard the fact, however, that presently -- and probably during the decade of the 1960's -- we have and will arrive at only the beginning of the introductory chapter of space research; the already-realized achievements, though magnificent, and the outlined problems of near future, though inspiring, do not yet represent real astronautics, literally, space travel. These are merely minor scientific exploratory ventures, the magnificent results of which will serve as a foundation for yet more magnificent achievements in the future.

In the present and future tasks of space research, the astronomers have only a secondary role, at least in the long-term planning of the ventures. However, at the moment, when we wish to arrive safely through space to the moon and the two neighboring planets -- in our view this is the very basic problem of the space research of this century -- astronomy will not be theoretical anymore, but will be converted into a considerably practical science, almost astrotechnics, and the cooperation on the part of astronomers will need to be much more basic (and flexible) than that of present time.

As much as we can see from previous investigations, it will be characteristic for future tasks of a really astronautical character that a new component will bear much more significance than it does at the present: namely, the useful constellation, that is, the mutual space position of a home planet (the earth) and a destination planet (e.g. Venus, Moon, Mars). It is also obvious from the studies that exploratory space ventures can be achieved only at the cost of enormous time losses, at least for the time being, while we aim at solutions of minimum energy. The starting schedule of the space vehicles will also be influenced by what we want to measure and whether we want to place the perigeum point of the course for a longer period on the semiglobe with day time, or on the semiglobe with night time. With regard to the previously mentioned space constellation, when starting a moon rocket, the period within which the success of the start has any chance will decrease to approximately a quarter of an hour. Even in case of an earth-to-moon trip, it might be necessary to wait for a few hours or days, especially if we are restricted from the viewpoint of the starting place, the time of starting and arrival, or f.e. illumination circumstances. This is valid to an increased extent for a Mars or Venus expedition. In this case, only a few days of the year are suitable for starting; after arriving at the destination planet there is a longer waiting period needed for the space ship to reach again

the favorable constellation required for the return.

We might state that in real space travel these time losses will determine the feasible or unfeasible character of the whole venture. This means that we have to alter our ideas on time; thus, in light of the mutual relationship between the course to be traveled and the time, our design concepts relating to velocity and consequently to acceleration are to be considered differently, than those we have been accustomed to during previous space experimenting.

This will result primarily in choosing the scale of the maximum and practical accelerations differently than that of former experiments. It is worthwhile to mention that an astronautic carrier rocket starts generally with an acceleration of a = 1.1-2 g; then, the acceleration suddenly increases due to the consumption of the propellant. Supposing a mass ratio 6:1 between initial and final masses of a given rocket stage, then -- assuming constant thrust for the rocket engine -- the final acceleration would be six times the initial one. Therefore, it is not surprising that present day astronautical carrier rockets have a final acceleration of 4-6 g in the first stage and 30-40 g in the final stage. The latter facts are to be specially considered when we are concerned with manned rockets or satellites.

We shall not forget that in the case of present day rockets the entire acceleration period lasts for a few minutes at the most; this is followed by flight on a gravitational trajectory. This solution suits the endeavors and conditions aimed at minimum energy.

However, at the moment, when the space travelers have ample time to achieve acceleration (in case of a real space-exploring venture), the former statements are not valid any more and the value of allowable and practical acceleration can and shall be selected in an entire different manner.

Table 1

Final velocity versus acceleration and accelerating time

$v_e = f\ (t, a)$ km/s

Time t	10 m/s² ~g	1 m/s² ~0,1g	0,1 m/s² ~0,001 g	0,01 m/s² ~0,001 g	0,001 m/s² ~0,0001 g
1 day	864	86	8,6	0,86	0,086
1 week	6,000	600	60	6	0.6
1 month	26,000	2,600	260	26	2.6
1 year	c	30,000	3,000	300	30

We have given the final velocities v_v (in km/sec) in Table 1 which, depending on the acceleration, will be reached at the end of the indicated accelerating period. As it is shown, in case of an acceleration of 1 g and an acclerating time of one year, even the velocity of light could be reached theoretically, if relativistic relationships are neglected. We will not need this, however, for a long time.

Until we are engaged in tasks of an astronautic character as mentioned before, there will not be necessary a total velocity increase bigger than Δv-30 ~ 60 km/sec, which would meet the requirements for reaching the escape veolocity, establishing a stable circular orbit around the destination planet, performing occasional landings and take-offs, and furthermore, returning into the atmosphere of the earth. By means of Table 1 the magnitude of effective accelerations to be applied according to the allowable accelerating time may be determined. This is shown in Table 2, where we have summarized the accelerating times for a total velocity increase of Δv =30 km/sec (that is, for the case of a Moon expedition with one landing or a Mars expedition without landing).

Table 2

Accelerating times pertaining to total velocity increase

Δv = 30 km/sec.

Acceleration, a	Accelerating time
g	few minutes, for special purposes, e.g. starting, returning
0.1 g	few hours
0.01 g	few days
0.001 g	one month
0.0001 g	one year

We can see that inasmuch as accelerating times with an order of magnitude of a month and a year are allowed, the value of the effective acceleration does not need be more than one thousandth and one ten-thousandth g, respectively. This fact

puts the new electric propelling mechanism in an entirely different light and calls attention to the fact that we can and must calculate with thrust to mass and thrust to weight ratios, wholly different to those we have been accustumed to with our present day rockets.

We may talk about the crisis of the chemical rocket, whereas I_{sp} =400 sec (=kpsec/kp) is the practical upper limit which may ever be reached by means of not the present day, but the contingently best future rocket fuels. We cannot expect an energy liberation of more than approximately 4 kcal/g from chemical reactions. In this field only the introduction of so-called free radicals (e.g., use of atomic hydrogen) could bring some improvement, but for the time being, sounding experiments only are being done in order to solve the stabilization of the extraordinarily active free radicals (e.g., by means of deep cooling).

The question of using atomic energy also arises. The atomic reactor or any other type of the controlled nuclear reactions represent unlimited heat sources.

The main difficulty arises from the fact that the propelling substance itself, the propellant, that is, the gas or steam discharged by the rocket engine, is simply unable to pick up energy of more than a certain quantity. By increasing the quantity of the introduced energy, the translational, rotational and vibrational energy levels become gradually saturated; first dissociation, later ionization occur. After reaching this limit condition, the substance is practically not only incapable of picking up more energy, but every state change results in more or less of an energy loss. After arriving at a state completely ionized, the substance is incapable of taking on more heat energy; thus, it is in vain to try introducing newer energy quantities, since the value of the discharge velocity w and the specific impulse Isp respectively cannot be increased. At this limit condition the thermal rocket becomes "choked up" as a driving engine and is unable to provide excess power for further acceleration.

Since during this process the substance became ionized, it appears to be an obvious solution that we look for some sort of electrical acceleration instead of increasing the temperature (that is, increasing specific heat energy) in order to accelerate the propellant. Modern atomic physics has perfected these accelerations methods to a large extent; we may find a particle accelerator even in television tubes.

The above concepts lead logically to various types of electrical drive, such as the ion rocket engine.

3) The Structure of the Ion Rocket Engine

The Structure of an ion rocket engine is shown in Figure 1. The suitably-selected propellant arrives from the propellant tank through the feeding system at the ion generator with a precisely - determined power. From the ion generator, the ions enter the ion accelerating chamber (the actual ion rocket engine), from where they are discharged, accelerated in the form of a high-velocity ion stream. The reaction force produced by the discharged ion stream provides for the thrust. At the same time, electrons are also liberated in the ion generator.

There is a proper energy source (e.g. an atomic reactor) necessary to feed the entire engine, and some sort of thermomechanical energy-converting apparatus (e.g. a steam or gas turbine) to transform the heat of the energy source into power driving the electrical generator. This generator uses up excessive electrons from the ion generator and feeds them into the electron accelerating chamber, from where they are discharged, accelerated in form of an electron stream supplying part of the thrust. It shall be also noted that significant heat losses, unavoidable in turbines, occur; they are to be dissipated by adequate cooling equipment. The indispensabley important part of an ion rocket engine is the chamber accelerating and discharging the electrons. Lacking this chamber, the apparatus, permanently loosing positive ions, would soon become negatively charged to such an extent that its further operation would be impossible due to polarization effects; the ion rocket engine would break down. It is, therefore, important that the electrically neutral state of the whole rocket engine and thereby the rocket itself be preserved and continously re-established.

It is not the purpose of our study, to review all the technical problems related to ion rocket engines. Therefore, here we deal merely with aspects which are primarily physical, and do not examine purely technical problems pertaining to f.e. electrical energy source, cooling, etc.

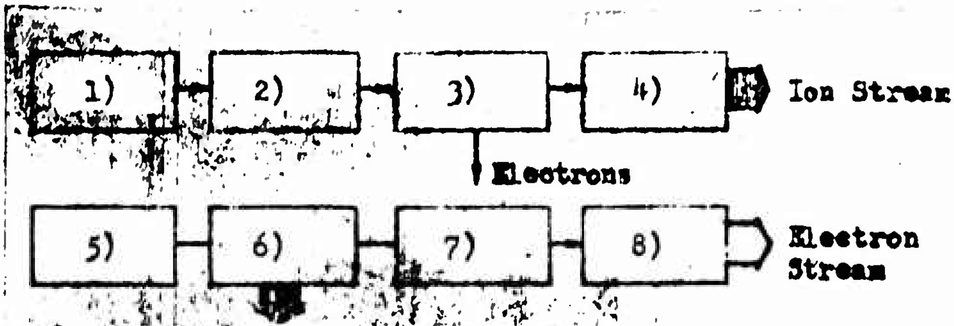


Figure 1.
Sketch of the Structure of a Typical Ion Rocket Engine.

1) Propellant Storage
2) Propellant Feeder
3) Ion Generator (Anode)
4) Ion Accelerator
5) Energy Source
6) Thermo-Mechanical Converter
7) Electrical Generator
8) Electron Accelerator

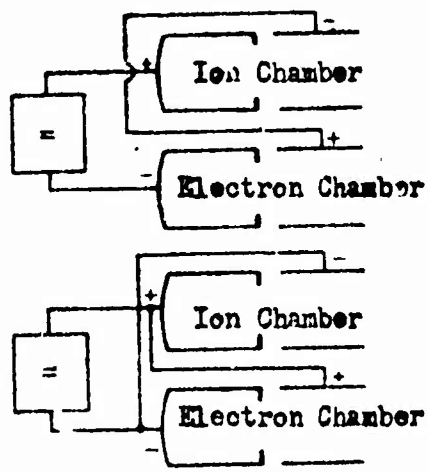


Figure 2.
Two basic types of ion rocket engines. Above is the series connection; below, the parallel-connected system.

4) Basic Equations of the Ion Rocket Engine

For an evaluation of any jet propulsion engine, that is, a reaction engine operating on the loss of materials discharged in the direction opposite to that of movement, the thrust is the most important charactoristic. If the propellant utilized in the engine is on the "vehicle" itself, then in accordance with Newton's second and third laws the equation of the thrust is as follows:

$$Ma = P = \sum n_i . m_i . w_i \qquad (3)$$

where P is the thrust (in dyns), M is the mass of the apparatus (gram), a is the accoleration of the vehicle (cm/sec2), n_i is the number of discharged particles per second, w_i is the relative velocity of the particles referred to the vehicle (cm/sec), and m_i is the mass of the particles (gram).

The quantities of equation (3), that is, the value of n_i, m_i and w_i, depend upon the selected propellant and the accelerating apparatus of the engine.

The simplest conception of the ion is to consider it as an atom with its electron shell deprived of one or more electrons. In this case, the positive charge of the nucleus prevails and determined the electrical behavior of the whole particle. For the sake of simplicity, we consider in our study only single-ionized substance; thus, we asume that the atom has lost only one electron. This assumption is in accordance with the line followed in present day experiments (cesium-ion rocket), but not necessarily acceptable for the propellant to be used in future ion rocket engines.

If we are after the ion mass of a substance with atomic weight A, this can be readily determined by knowing the Avogadro number and the atomic weight (atomic mass number), since the Avogadro number renders the number of particles in one gram-atom or gram-molecule of the substance, respectively. Thus, the mass of one ion of the propellant with an atomic weight A is

$$m_i = \frac{A}{6.0248 . 10^{23}} - m_e \text{gram.} \qquad (4)$$

By means of equations (3) and (4) the thrust of the ion rocket engine may be expressed

$$P = M.a = \frac{A . n_i . w_i}{6.0248 . 10^{23}} + m_e (n_e . w_e - n_i . w_i). \qquad (5)$$

where n_i and n_e are numbers of the ions and the electrons per second respectively, $6.0248 \cdot 10^{23}$ is the so-called Avogadro number, m_e is the mass of an electron ($9.1084 \cdot 10^{-28}$g), and w_i and w_e are the velocities of the ions and the electrons, respectively (cm/sec).

Although the electrical charge of the particles discharged from the ion rocket, that is, the charge loss of the apparatus per unit of time, influences the electrodynamic behavior of the whole rocket (due to induced charges on metal surfaces and subsequent electrical repulsive forces of various kind), this problem will not be dealt with here, though the more accurate calculation of the correction requirements resulting from such secondary effects would undoubtedly be interesting in the future.

If we take, however, into consideration the fact that the election mass is negligibly small as compared with that of the ion, and that it, therefore, is extremely unlikely (even technically impractical) to increase the velocity of the electrons in order to increase the thrust, then the following simpler formula may be obtained for the thrust of the ion rocket:

$$P = \frac{A \cdot n_i \cdot w_i}{6.0248 \cdot 10^{23}} \text{ dyn.} \qquad (6)$$

The number of particles is obviously linearly proportional with the electrical current, since

$$n = \frac{3.10^9 \cdot i}{e} = 6.246.10^{18} \cdot i, \qquad (7)$$

where e is the charge of an electron, that is, $4.8029 \cdot 10^9$ electrostatic units, and i is the current. The above equation does not contain any more the coefficient 3.10^9 (more accurately, $2,9979 \cdot 10^9$), which is necessary in order to convert the current i given in amperes from technical units into CGSE units.

The velocity of the discharged particles results from the change in kinetic energy; this kinetic energy increase is the effect of the electrostatic field.

If we assume that the particles started from rest, with zero initial velocity, their kinetic energy is given by following formula:

$$\frac{m_i \cdot w_i^2}{2} = \frac{e \cdot V}{300} \text{ erg;} \qquad (8)$$

hence, we may express the discharge velocity

$$w_i = \sqrt{\frac{e \cdot V}{150\, m_i}} \quad \text{cm/sec}, \tag{9}$$

where V is the potential difference accelerating the particles, m_i is the mass of the ion, and factor 300 is again the conversion factor between CGSE and the practical unit systems.

By substituting the explicit term for the ion mass into former expression, the following formula is given for the discharge velocity of the ions:

$$w_i = 1.389 \cdot 10^6 \sqrt{\frac{V}{A}} \quad \text{cm/sec} \tag{10}$$

In order to verify that neglecting the mass and velocity of the electrons does not affect the dynamic relationship of the ion rocket engine, let us mention here that the discharge velocity of an ion with its mass decreased by the mass of an electron may be given as follows:

$$w_i = \frac{1.389 \cdot 10 \sqrt{\frac{V}{A}}}{1 - \sqrt{\frac{1}{1836\, A}}} \tag{11}$$

where the quantity under square root in the nominator is justified to be taken as one. Thus we arrived at formula (10).

The discharge velocity of the electrons is given by the following formula:

$$w_e = 5.952 \cdot 10^7 \sqrt{V_o} \quad \text{cm/sec.} \tag{12}$$

Finally, the thrust expressed in the values of the accelerating potential and the current can be written as follows:

$$P = 14.39 \cdot i_i \sqrt{A \cdot V_i} \quad \text{dyn}, \tag{13}$$

where we may again neglect the effect of electrons on the thrust.

As a matter of fact, the negligibly small thrust of the electron is an "unavoidable burden" which we need in order to maintain the electrically neutral condition.

In practise, it is necessary to have a possibly equal number of positive and negative charges discharged in unit time. This condition is equivalent to having the same current flowing in both the ion accelerating and the electron accelerating chambers. In this case also the ratio of the voltages can be determined.

$$\frac{V_e}{V_i} = \frac{m_e}{m_i} \approx \frac{1}{1836 \cdot A} \quad (14)$$

This value shall be precisely established when determining the power of the propelling stream.

Since this study presumes ideal conditions, in our discussion the energy requirement is meant to be the net power requirement which is to be introduced into the single accelerating chambers; this power is not identical at all with the output power of the electrical apparatus.

When we transform electrical energy into kinetic, the efficiency of this action depends upon what type of particle accelerators we use, how successful the making of the propelling ion stream a practically parallel stream is, what losses (ohmic, inductive, etc.) occur in the transmission system, and what additional losses are caused by the collision of the particles with structural elements of the engine (mainly with the wall of the accelerating chambers).

The net accelerating power of the ion stream discharged from the chambers of the ion rocket engine is as follows:

$$N = \left[\frac{n_i \cdot m_i \cdot w_i^2}{2} = \frac{n_e \cdot m_e \cdot w_e^2}{2}\right] \cdot 10^{-7} \text{watt} \quad (15)$$

where the first term gives the power of the ion current and the second term gives that of the electron current. The question again arises whether the second term cannot be easily neglected. For the time being, we shall answer "yes" to this question.

The value of the actual power may be determined, depending upon what electrical connection we may employ for the ion accelerating and electron accelerating chambers. As shown on Figure 2, there are two solutions possible.

[redacted]

The first solution is the so-called series connection, where we maintain the voltages so that the velocities of the ions and electrons are practically identical. We may again neglect the electron participation in the drive power; and thus, we get

$$N_s = \frac{n_1 . A . w_1^2}{12.0496 \,.\, 10^{30}} \text{ watt.} \tag{16}$$

If the accelerating apparatus is of parallel connection, then the power is given by the following formula:

$$N_p = \frac{n_1 . A . w_1^2}{6.0248 \,.\, 10^{30}} \text{ watt.} \tag{17}$$

This formula appears to prove that we have a bigger (twice as much) drive power in this case. This is, however, a false conclusion, because in this case the accelerating voltage of the electrons should be increased to an extraordinary extent. Since the accelerating voltage determines mainly the electrical power requirement of the accelerating apparatus, we quickly arrive at an economical limitation. According to present day investigations, the series connection proves better in every case.

5) More Important Relationships for the Ion Rocket Engine

From the above discussed relationships, we can derive the calculation characteristics, by means of which the ion rocket engine may be compared to the classical rocket, and some technical design data may be established. When the most important characteristics are investigated, the following data may be defined:

a) specific power referred to unit weight of the (entire) rocket,

b) specific thrust determined by the maximum reaction force resulting from 1 kw input power,

c) specific propellant consumption

d) specific impulse, that is, thrust expressed in kp, which is gained from consumption of k kp/sec propellant,

e) mass ratio, etc.

All these characteristics are in fact defined by the factors as follows:

a) the weight of the rocket

b) the specific thrust of the rocket (thrust per weight or thrust per mass, respectively)

c) the accelerating potential

d) the atomic weight of the propellant as quantitative characteristic of the substance.

When we look for analogy with the classical chemical rockets, we find that the accelerating voltage approximately corresponds with the temperature in the engine and the atomic weight with pressure in the combustion area of the engine.

First, let us express the power of the propelling stream in terms of mass, acceleration, and discharge velocity:

$$N = \frac{1}{2} M a w_1 . 10^{-7} \text{ watt.} \qquad (18)$$

We can again here neglect the power of the electron current or, in case of detail analysis, we may take it into account by putting a correction term into the power expression for the ion.

If the weight of the rocket is

$$G = M . g \qquad (19)$$

and the accelerating thrust is

$$P = M . a, \qquad (20)$$

then obviously,

$$\frac{P}{G} = \frac{a}{g} = \alpha. \qquad (21)$$

where the latter coefficient gives the ratio between the effective acceleration and g (now we refer again to Points 1 and 2).

Taking former results (including factor α) into account, we arrive at formulas (22) and 23):

$$\frac{N}{M} = \frac{a \cdot w_1 \cdot 10^{-7}}{2} \qquad (22)$$

where the power is given in watt/gram, or

$$\frac{N}{P} = \frac{\alpha \cdot w_1}{2 \,.\, 736} \ \text{hp/kp}, \qquad (23)$$

respectively, where the specific power is expressed in hp/kp.

The same ratio can be determined by means of accelerating voltage, namely, if the same current is flowing through both accelerating chambers (in case of series connection), the formulas will be as follows:

$$\frac{N}{M} = 0.0695 \ a \sqrt{\frac{V}{A}} \ \text{W/g} \qquad (24)$$

$$\frac{N}{P} = 91.5 \sqrt{\frac{V}{A}} \ \text{hp/kp} \qquad (25)$$

From theoretical investigation it is clear that should the velocity of the ions and electrons be the same, then the correction factor is equal to one and the power to weight ratio is minimum. If we apply, however, the same accelerating voltage in both systems, then the power to weight ratio is double, furthermore, only the rocket with series connection is feasible. This alternative is the more favorable one from operational and other viewpoints It is also to be noted that in case of a series arrangement, the currents are smaller; thus, the pertinent iron and copper losses are also smaller.

The bigger the thrust to weight ratio is, the bigger the power to weight ratio will be; this is understandable, since bigger thrust is achievable only at the expense of feeding in greater power. Moreover, the square root of the ratio of the acceleration voltage to propellant atomic weight also has a part in the above equation; with other data on propellants

thought of as being generally useful in ion rockets, this is shown in Table 3.

Table 3

Some Important Characteristics of Ion Rocket Propellants.

Accelerating voltage is V = 12,000 V.

Substance	Atomic weight, A	V/A	A, V/A	i_1/P_{kp} A/kp
Hydrogen (H)	1.008	12,000	110	620
Cesium (Cs)	132.91	90	1265	54
Uranium (U)	238.17	50.2	1645	41.4

Since, in conformity with formula (24), the power to weight ratio is proportional to quotient V/A, it is therefore, worthwhile using elements of heavier atomic weights as a propelling substance. The ions of uranium U-238 (in single-ionized state) f.e. could produce theoretically a 15.4 times greater thrust than that of a hydrogen propellant.

We have already determined the value of the thrust. Now, using the terms of the accelerating power and the accelerating voltage, we arrive at the following formula:

$$P = 14.39 \cdot N \cdot \sqrt{\frac{A}{V_1}}\ \text{dyn.} \qquad (26)$$

If we relate the thrust to the unit power again, the form of the formula becomes simplified as follows:

$$\frac{P}{N} = 14.39 \sqrt{\frac{A}{V_1}}\ \text{dyn/W} \qquad (27)$$

and

$$\frac{P}{N} = 0.0108 \sqrt{\frac{A}{V_i}} \text{ kp/hp.} \tag{28}$$

The reciprocal value of this expression is interesting also, since it renders the quantity of power in watts to be introduced in order to achieve one dyn or one kp thrust. The formula will be:

$$\frac{N}{P} = 0.0695 \sqrt{\frac{V}{A}} \text{ W/dyn.} \tag{29}$$

It is also clear from this equality that the power consumption increases with the increasing ratio of the accelerating voltage to the atomic weight of the propellant.

This fact leads us to the conclusion that the value of the accelerating voltage has an economical maximum which would be hardly worthwhile to exceed.

We have earlier derived -- in formulas (10) and (12) -- the discharge velocities of the ions and the electrons. These are:

$$w_i = 1.389 \cdot 10^6 \sqrt{\frac{V}{A}} \text{ cm/sec} \tag{10}$$

or

$$w_i = 13.89 \cdot \sqrt{\frac{V}{A}} \text{ km/sec} \tag{30}$$

for the ion streams (Figure 3), and

$$w_e = 5.952 \cdot 10^7 \sqrt{V_e} \text{ cm/sec} \tag{12}$$

or

$$w_e = 595.2 \sqrt{V_e} \text{ km/sec} \tag{31}$$

for electrons.

We have to determine also the output current of the ion rocket engine. In case of the series connection, the current is flowing through both the ion-accelerating and the electron-accelerating chambers; its value is obviously

$$i = \frac{N}{V_i + V_e} . \qquad (32)$$

If all the relating quantities are substituted and the electron accelerating chamber is again neglected, then the following formula may be written for the current (in amperes):

$$i_i = \frac{N}{V_i} \quad \frac{0.06959\ P}{A \sqrt{V/A}} . \qquad (33)$$

The above formula is to be slightly modified if the velocity of the ions and the electrons are different.

In case of parallel connection, the accelerating chambers require twice as much current and power as those of series connection. At the same time, however, the thrust gain achieved by means of the electrons is only 1/43 part of the thrust of the ion stream.

If we examine the ion current of an ion rocket engine, then from the preceding formulas it is clear that for a given thrust the value of the output current will change with an order of magnitude of 2, when uranium instead of hydrogen is used as propellant. At the same time, the use of a heavier propelling substance also offers significant technical advantages.

Let us now calculate the propellant consumption per second. To do so, we have to know partly the ion current and partly the atomic weight of the propellant. Namely,

$$q_h = n_i . m_i . n_e . m_e \approx \frac{3 . 10^9 i_i . A}{6.0248 . 10^{23} . e} \qquad (34)$$

In this formula q_h is the propellant consumption per second (g/sec, may be kp/sec), i_i is the current flowing through the ion chamber (amperes), and e is again the charge of the electron. The consumption expressed in CGSE units is as follows:

$$q_h = 1.038 \cdot 10^{-5} \cdot i_1 \cdot A \ g/sec. \tag{35}$$

If we return to the values of the take-off weight and the specific acceleration again, then we arrive at the following two formulas:

$$q_h = 7.2 \cdot 10^{-7} \cdot P \sqrt{\frac{A}{V}} \quad g/sec, \tag{36}$$

$$q_s = 7.06 \cdot 10^{-4} \, G_{t-o} \cdot \alpha \sqrt{\frac{A}{V}} \,. \tag{37}$$

After knowing the thrust and the propellant weight consumed per second, we may calculate also the specific impulse of the propellant of the ion rocket engine. As a matter of fact, here the specific impulse of not only the propelling substance, but that of the entire engine is in question; accordingly, it characterizes the quality of the whole apparatus. By means of the formulas derived above, we may write:

$$I_{1i} = \frac{P}{q_h} = 1417 \sqrt{\frac{V}{A}} \quad kpsec/kp. \tag{38}$$

Similarly for the electrons,

$$I_{1e} = 6.072 \cdot 10^4 \sqrt{V_e} \quad kpsec/kp, \tag{39}$$

*which may be again neglected.

The above formulas make it clear that by means of the ion rocket engine a very significant specific impulse can be achieved (the reader should remember the ideal value I_{sp} - 400 sec of the best chemical propellants). If the value of V/A is 100 (which is the roughly-approximated case of 13 kw of accelerating voltage when cesium is used as propellant), then the specific impulse is 14,170 sec,, enormous value. If it were possible to increase further the accelerating voltage, we may get even larger values. It shall be noted, however, that the occurrent technical difficulties are too big, lest be the accelerating voltage of 15 kv exceeded (Figure 4).

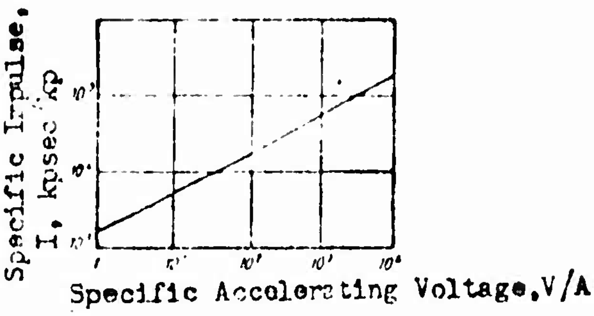
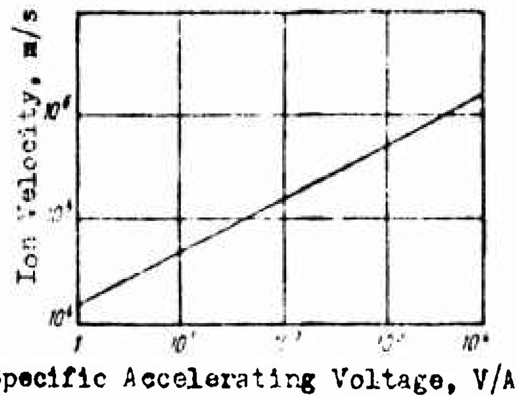


Figure 4.

Specific Impulse versus Specific Accelerating Voltage. Initial Value is Approximately 1400 kpsec/kp. (Greatest chemical impulse is approximately 400 kpsec/kp.

Figure 3.

Discharge Velocity of Ions versus Specific Accelerating Voltage (=accelerating voltage/atomic weight). Initial Value is approximately 14 km/sec.



The current flowing through the ion chamber is extraordinarily large. It is of an order of magnitude of thousands, even ten thousands, of amperes; correspondingly, the required electrical power (the product of current and accelerating voltage) can be expressed only in megawatts. This is one of the fundamental problems in technically realizing the ion rocket.

By means of certain simplifying assumptions mainly aiming at the formation of accelerating electrodes, uniformity of their fields, etc., the density of the particle current may be given as follows:

$$3 \cdot 10^9 \cdot \sigma = \frac{1}{9\pi d^2} \left(\frac{2e}{m_1}\right)^{\frac{1}{2}} \left(\frac{V}{300}\right)^{\frac{3}{2}} \quad . \qquad (40)$$

Here is the current density (amp/cm^2) e and m_1 are the charge and the mass of the particle, respectively; V is the accelerating voltage (volt), and d is the distance between accelerating electrodes. After introducing the quotient V/A again, the formula for current density will be of the following form:

$$\sigma = 0.547 \cdot 10^{-7} \frac{A}{d^2} \left(\frac{V}{A}\right)^{\frac{3}{2}} \quad A/cm^2. \qquad (41)$$

The quotient of the ion current and the ion current density gives the cross-section of the ion chamber:

$$F = \frac{i}{\sigma} = 0.127 \cdot 10^{-7} \cdot P \cdot \left(\frac{d}{V}\right)^2 \quad cm^2. \qquad (42)$$

Similarly, we can determine also the current density of the electron-accelerating chamber and the cross-section ratio for the two chambers.

We must also mention the structural difficulties: we can not give the accelerating electrodes the shape of a flat plate, but that of a ring screen. This has its own difficulties, since in the case of extraordinarily big currents, the electromechanical destructive effects of the particles colliding with the screen is unimaginably big. In addition, there is also extremely great heat caused by the collision.

From formula (42) it is clear that the quotient V/d, that is, the intensity of the electric field, will also influence the structure of the ion chamber. The aim is to increase the electric field to the limit of feasibility. It should be added

that in the electron accelerating chamber, we may have electrode distances significantly shorter than those of the ion accelerating chamber; thus, by taking all the previous relationships into consideration it is possible to have dimensions of the same order of magnitude for the electron accelerating chamber, as well as the ion accelerating chamber.

6) Some Practical Data.

The ion rocket engines have been worked on for approximately a decade. Since -- in conformity with the thoughts discussed hereunto -- the theoretical background of these propelling engines may be clarified by not too complicated relationships, the first experiments have been recently performed.

It is unfortunate that the results of the experiments performed on ion propelling engines are -- though understandably -- still kept in secrecy.

In spite of the fact that the ion rocket engines are in all probability unsuitable to perform military tasks, the corresponding development in the USA is financed entirely by the military budget. In the Soviet Union, the investigation of the question is recognized as partly a fundamental research matter, but it is obvious that it does not intend to issue detailed information until all the international legal problems of space research are adequately solved.

It is known that experimental ion rocket engines producing 0.5 - 1.0 kg thrust are already operating in laboratories. They require a considerable amount of electrical power; thus, the problems pertaining to the investigations are mainly represented by introducing electrical energy, consuming it, etc.

We also should not forget the fact that maintenance and control over the relatively high (12-15kv) accelerating voltages and the hundreds of amperes pertinent to the above thrust values is a fairly complex tasks requiring special equipment generally similar to that employed in short-circuit investigations of high voltage systems.

The essential requirement for the propelling engine is stable, continuous operation. This is, however, dependent to a large extent on the accelerating voltage. In case of a low accelerating voltage, the current of the ion stream is relatively small, and a considerable number of the ions suffer collision. Proportional to the increasing voltage, the current also increases and soon reaches the limit value established by the capacity of the ion source. The biggest current may be measured at approximately 5,000 V accelerating voltage. By further increasing the voltage, the current tends to decrease and has its minimum

at approximately 12,000 V. Somewhat higher voltages than this should be maintained for stable operation, whereby the current is acceptably small. The latter can be further decreased by adequately shielding and cooling the accelerating electrodes. In addition, the electrodes should be so located that the divergence of the outflowing ion stream is the smaller, thereby, the damaging collision losses are prevented. From this point of view, the ion stream strongly resembles the outflowing fluid stream at the discharge orifice of the classical chemical rocket.

We know about one datum reported on the value of the accelerating electric field intensity: the minimum allowable value is $3 \cdot 10^4$ V/cm.

We may regard the following data as the most important three characteristics of the ion rocket engine:

1) The ratio of the accelerating voltage to the effective atomic weight (may be molecular weight) of the propelling substance; that is the quotient V/A;

2) The entire weight of the ion rocket, M_o or G_o respectively;

3) Finally, tge ratio between the thrust and the rocket weight.

The design datum on latter factor is around 10^{-3}, preferably 10^{-4}.

The most practical propellant may be cesium, rubidium, and perhaps uranium. Although information thus far reported deals exclusively with cesium (which seems to be justified even by the relatively small ionization potential of this metal), the investigation may not exclude f.e. the mercury or the uranium; it would be worthwhile to also examine other similar metals.

The actual design of the ion rocket engine -- like any technical task -- leads to a search for compromise. We have to find the most practical and favorable ratio between the voltage accelerating the propellant, the thrust; and the weight of the whole mechanism. Thereby we may have optimum conditions for the specific power, the dimensions, the weight, and the operating time. We can derive useful relationships by utilizing the formulas of classical rocket mechanics, since we know that the velocity change of the rocket (with gravitational losses neglected) may be given by the so-called rocket formula of relatively simple form:

$$\Delta v = w \cdot \ln \frac{M_o}{M_1} \cdot \qquad (43)$$

When expressing in exponential terms:

$$\frac{M_o}{M_1} = e^{\frac{\Delta v}{w}}, \qquad (44)$$

we get the ratio of the take-off weight to the weight of the empty rocket, which can be expressed by means of the specific thrust factor α (specific acceleration) and which appeared previously in our discussion many times (the ratio V/A and the operating time t of the ion rocket engine). We get the following formula:

$$\frac{M_o}{M_1} = \frac{M_o}{M_o - q_h t} = \frac{1}{1 - 7.06 \cdot 10^{-4} \cdot \alpha \cdot \sqrt{\frac{A}{V}} \cdot t} \qquad (45)$$

From the previous formula we may calculate also the operating time of the engine. This will vary according to how big a specific thrust is required (the smaller the specific thrust, the longer the operating time); furthermore, it is longer if the accelerating voltage is increased. Shorter operating time is necessary, however, if a propellant with bigger atomic weight is employed.

From our previous discussion, there arises one basic problem, which is not answered here. However, the ion rocket engine mounted on a rocket will be feasible only when we have succeeded in finding electrical sources with adequately large specific powers. Only by means of such power sources can we satisfy the considerable energy requirement of the ion engine.

7) Examples

The significance of the application of the ion rocket engine may be demonstrated by a simple example according to Moeckel.

Let us assume that we want to send a space ship with 8 men in it for an exploration of Mars. The weight distribution of this space ship would be as follows:

Cabin and its equipment	20 t
equipment for the expedition (including auxiliary rockets, etc.)	25 t
food, oxygen, etc. (4.5 kg/person/day)	45 t
Total	90 t

If various propellants are now examined, then the starting weight of the space ship with useful weight of 90 t will be as follows:

1) Using a chemical propellant (I_{sp} = 300 sec), the initial weight is 3,000 t if the rocket structure represents only 5% of the total weight.

2) Using a chemical "super propellant" with a capacity of I_{sp} =400 sec, the initial weight of a rocket of similar structural quality is 900 t.

3) In the case of an atomic-driven thermal rocket with specific impulse I_{sp}=800 sec and thrust to weight ratio = 1, the initial weight of the rocket is 400 t, which might be decreased to 300 t by employing generators of more advanced construction.

4) Using an ion rocket engine with specific impulse I_{sp} =10,000 sec, the initial weight decreases to 180 t.

We shall not forget that first we have to have the space ship somehow orbited around the Earth. If we take into consideration the fact that for putting 1 kg into orbiting in an artificial moon trajectory there is a 50-200 kg carrier rocket needed, then we can see that the weight decrease achieved by employing an ion rocket engine might enable putting such a space ship into orbit. Thus, we may state that while the expedition using a chemical rocket is unfeasible according to the present stage of technics, using an ion rocket engine already approaches reality.

Following E. Stuhlinger, we have in Table 4 summarized the data on ion rockets employed to perform some typical space exploring tasks.

Table 4

Data of Spaceships Propelled by Ion Rocket Engines
(After Stuhlinger)

Task	A	B	C	D	E	F	G
Initial weight	5	72.2	136	435	5.5	10.9	16 tons
Useful weight	4.9	50	100	150	1	1	1 "
Propellant weight	0.06	6.7	17	192	2.8	5.9	8.8 "
Engine weight	0.04	15.5	19	93	1.5	4	6.2 "
Accelerating time	0.03	0.08	0.17	1.6	1.5	2.5	3 years
Power	5	4650	5700	27900	450	1200	1900 kw
Thrust	0.027	16.9	19.7	46.8	0.75	1.36	1.9 kp
Ion velocity	38	60	60	120	120	180	200 km/s
Final velocity	0.46	6	10.5	72	90	140	160 km/s
Initial acceleration	0.054	2.29	1.45	1.06	1.38	1.22	$1.16 \cdot 10^{-4}$ g
Voltage	1000	2325	2420	9950	10100	22200	28000 volt
Ion current	5	2000	2360	2800	44.5	54	68 ampere

Legend: A-Trajectory correction of the artificial moon, B - Trajectory modification of the artificial moon, C - Moon rocket, D - Mars rocket, E - Jupiter sounder, F - Saturn sounder, G - Interplanetary research sounder.

At the present time only the fundamental research is in progress in connection with ion rocket engines. The basic relationships previously discussed may be supplemented to a considerable extent and further developed toward solutions which are also technically more or less realistic. The few data reported heretofore permit concluding that these propelling engines might really become useful means of future space research. The time requirements of related developmental work are to be measured, however, rather in decades than years, with special regard to pertinent difficulties of technical, metallurgical, energetical, and other natures.

10,243

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